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Stationary Plasma Thruster Evaluation in Russia

Summary Report

John R. Brophy



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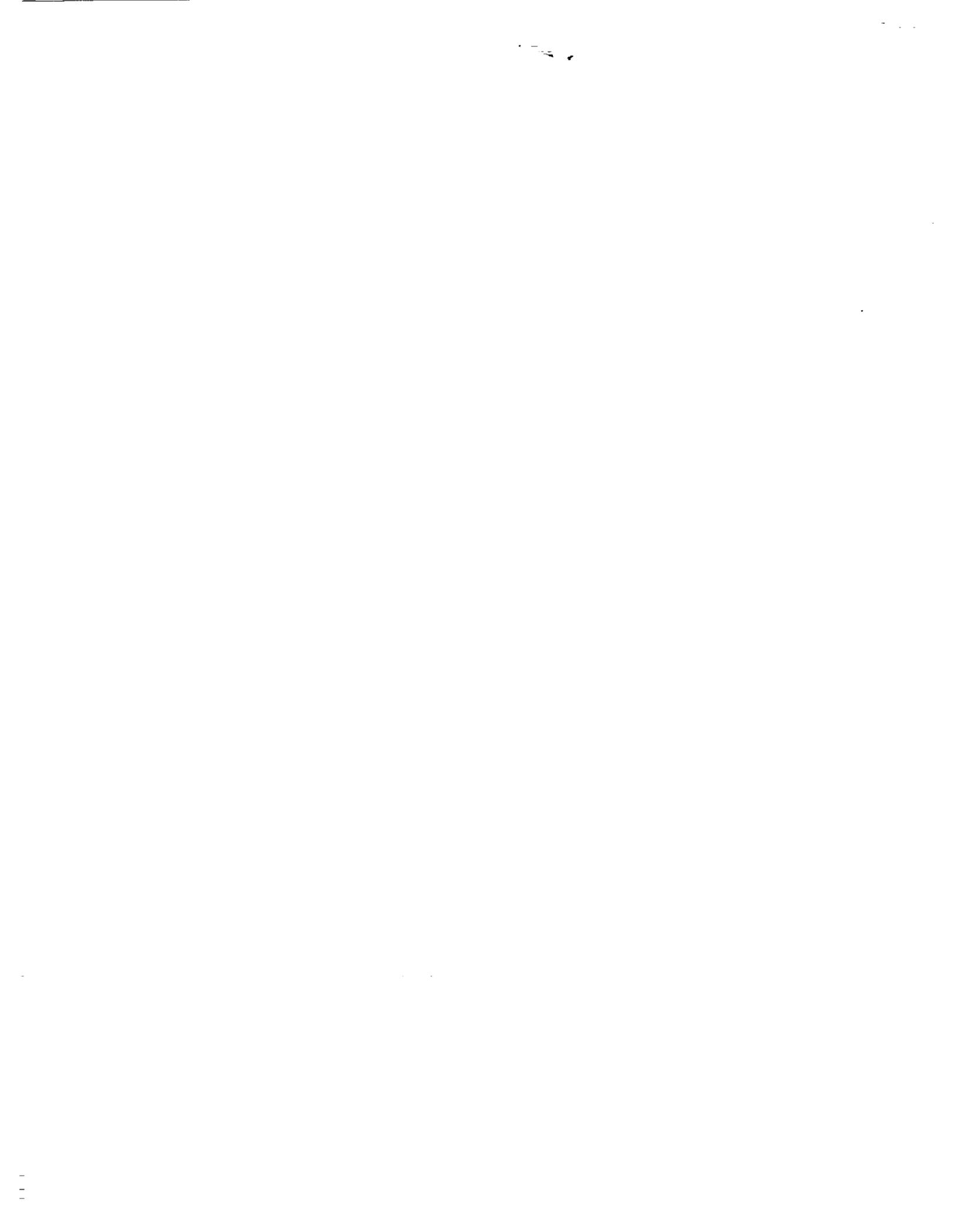
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Abstract

A team of electric propulsion specialists from U.S. government laboratories experimentally evaluated the performance of a 1.35-kW Stationary Plasma Thruster (SPT) at the Scientific-Research Institute of Thermal Processes in Moscow and at "Fakel" Enterprise in Kaliningrad, Russia. The evaluation was performed using a combination of U.S. and Russian instrumentation and indicated that the actual performance of the thruster appears to be close to the claimed performance. The claimed performance was a specific impulse of 16,000 m/s, an overall efficiency of 50%, and an input power of 1.35 kW, and is superior to the performance of western electric thrusters at this specific impulse. The unique performance capabilities of the stationary plasma thruster, along with claims that more than fifty of the 660-W thrusters have been flown in space on Russian spacecraft, attracted the interest of western spacecraft propulsion specialists. A two-phase program was initiated to evaluate the stationary plasma thruster performance and technology. The first phase of this program, to experimentally evaluate the performance of the thruster with U.S. instrumentation in Russia, is described in this report. The second phase objective is to determine the suitability of the stationary plasma thruster technology for use on western spacecraft. This will be accomplished by bringing stationary plasma thrusters to the U.S. for quantification of thruster erosion rates, measurements of the performance variation as a function of long-duration operation, quantification of the exhaust beam divergence angle, and determination of the non-propellant efflux from the thruster. These issues require quantification in order to maximize the probability for user application of the SPT technology and significantly increase the propulsion capabilities of U.S. spacecraft.

Acknowledgments

The author thanks the technical team members Dr. John W. Barnett (now with the Rocky Mountain Institute), Mr. David A. Barnhart (Air Force Phillips Laboratory), and Mr. John M. Sankovic (NASA Lewis Research Center), whose participation in the evaluation of the Stationary Plasma Thruster (SPT) was absolutely essential to the successful completion of this task.

The concept of evaluating the SPT in Russia using U.S. instrumentation stemmed from August 1990 discussions among Dr. Anatoly Koroteev (Director, Scientific-Research Institute of Thermal Processes in Moscow), Mr. Joseph Wetch (International Scientific Products in San Jose, CA), and Dr. Leonard Caveny (Strategic Defense Initiative Organization). Their efforts are helping to achieve and extend the lines of communication and governmental approvals.

The author thanks Dr. Victor Koba, Dr. Valary Petrosov, and Dr. Vladimir Brutky of the Scientific-Research Institute of Thermal Processes, and Mr. Alexander Bober and Mr. Boris Arkhipov of "Fakel" Enterprise for their efficiency, hospitality, openness, and friendship. The author also thanks Professor Alexis Morozov (Kurchatov Institute of Atomic Energy) and Dr. Vladimir Kim (Moscow Aviation Institute) for sharing their insights into the physics underlying the SPT.

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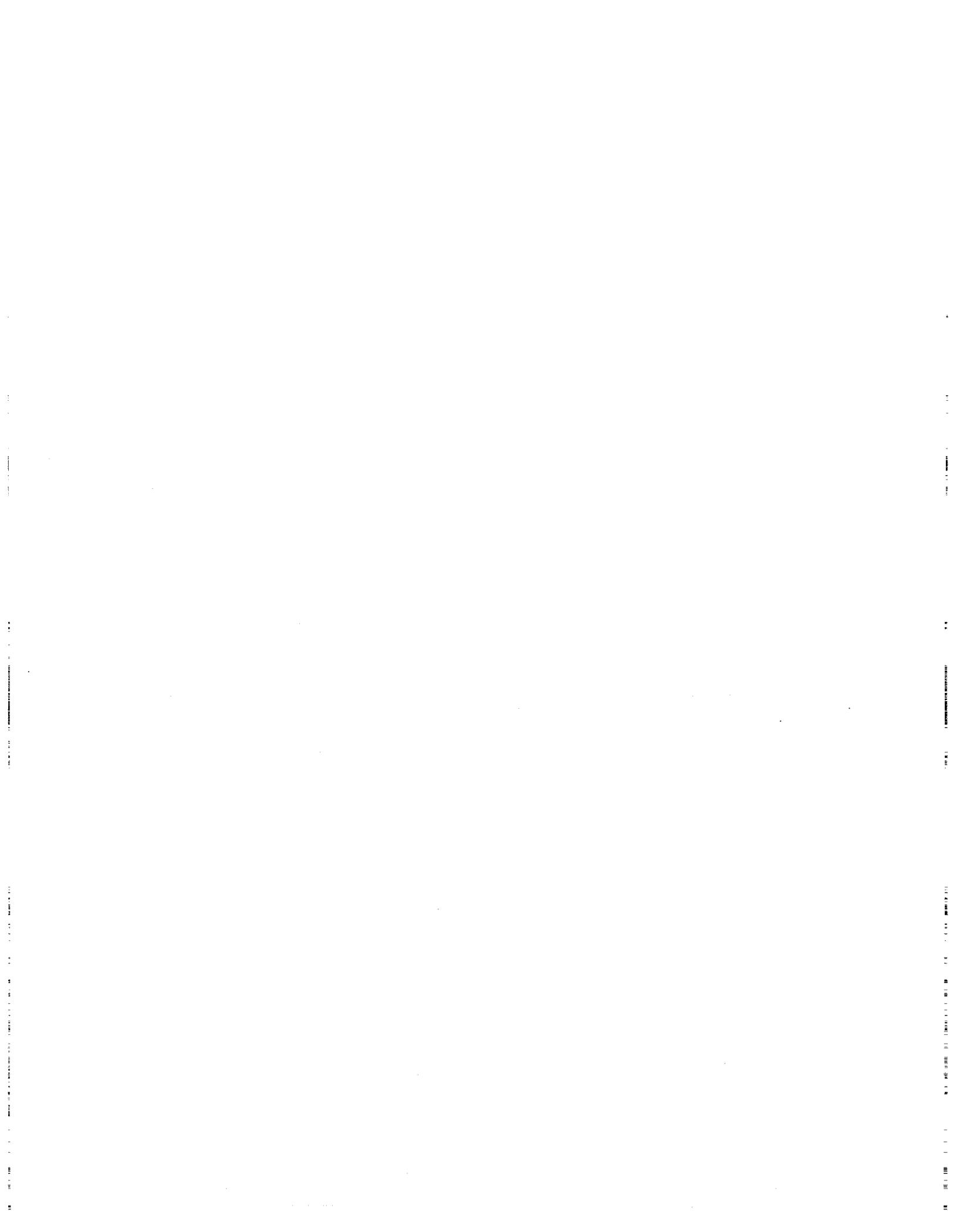
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1. Introduction

A team of electric propulsion specialists from U.S. government laboratories was formed to experimentally evaluate the performance of a Stationary Plasma Thruster (designated the SPT-100) in the Soviet Union over the past thirty years. The team traveled to two laboratories, the Scientific-Research Institute of Thermal Processes (NIITP) in Moscow and "Fakel" Enterprise in Kaliningrad (on the Baltic Sea) during September and October of 1991. Experimental measurements were made using U.S. instrumentation to assess the performance of the SPT-100 thruster.

The unique performance capabilities advertised for the SPT-100 electric thruster, together with the claims that more than fifty of the similar, but smaller, SPT-70 stationary plasma thrusters have been flown in space, initially attracted the interest of western spacecraft propulsion specialists. A two-phase program was initiated to evaluate the stationary plasma thruster performance and technology. The first phase of this program was to evaluate the SPT-100 performance in the Soviet Union using U.S. instrumentation. The results of this evaluation were to be used as the basis for deciding to proceed with the second phase of the program in which stationary plasma thrusters will be brought to the U.S. for additional experimental characterizations. This report summarizes the results of the phase I thruster performance evaluation performed in Russia and indicates that the performance of the SPT-100 appears to be close to its advertised performance and that the phase II activity is clearly worth pursuing.

The objective of the phase II activity will be to determine the suitability of the stationary plasma thruster technology for use on western spacecraft. This will include performance and endurance testing in U.S. electric propulsion test facilities and characterization of the charged particle and non-propellant effluxes from the thruster. These activities will assist the U.S. government and U.S. industries in assessing the desirability of pursuing the development of spacecraft propulsion systems based on stationary plasma thruster technology.

In the first western publications¹ giving SPT performance, the thruster was claimed to operate with a unique combination of specific impulse and efficiency. The claimed performance for the SPT-100 thruster is given in Table 1. Numerous studies (see for example Ref. 2) have shown that for Earth orbit raising and north/south stationkeeping applications of electric propulsion, the optimum specific impulse is in the range of 10,000 to 20,000 m/s (1000 to 2000 s).^{*} The SPT has been optimized to operate at a specific impulse of 16,000 m/s (1600 s) with xenon propellant. The results of this optimization have produced a thruster which is claimed to be 50% efficient at a specific impulse that is ideal for many near-Earth space applications.

^{*}The SI unit for specific impulse is meters/second, which corresponds to its definition as the ratio of thrust to the propellant mass flow rate. The electric propulsion community, however, often uses the related, but slightly different definition that the specific impulse is the ratio of thrust to the propellant weight flow rate (based on the value of the gravitational constant at the Earth's surface, $g = 9.8 \text{ m/s}^2$). In this case, the unit of specific impulse is seconds. The two definitions differ by the value of g . Definitions of other key terms are given in Appendix C.

Table 1. SPT-100 Performance Characteristics

PARAMETER	VALUE
Propellant	Xenon
Input Power	1.35 kW
Thrust	0.080 N (0.018 lb _f)
Specific Impulse	16,000 m/s (1600 s)
Efficiency	0.50
Xenon Flow Rate	5.0×10^{-6} kg/s
Lifetime	4000 hrs
No. of Start/Stop Cycles	3000
Thruster Mass	4 kg

The SPT-100 performance is compared to that of western electric thrusters in Figs. 1 and 2 for specific impulses up to 25,000 m/s (2500 s). The efficiency of electrothermal arcjets tends to decrease with increasing specific impulse. Furthermore, it is difficult to operate arcjets at specific impulses approaching 15,000 m/s (1500 s) due to thermal limitations of the expansion nozzle. In contrast, the efficiency of the ion engine decreases as the specific impulse is reduced below 25,000 m/s (2500 s) as a result of the increasing importance of ionization losses at low specific impulses. The performance of the "derated" ion engine from Ref. 3 has recently demonstrated performance which approaches that of the SPT-100 thruster; however, the "derated" ion engine is a factor of three larger in diameter than the SPT-100, a factor important for integration of the thrusters onto a volume-limited spacecraft.

Stationary plasma thrusters in sizes ranging from 50- to 290-mm diameter, with input powers ranging from 400 to 25,000 watts, have been tested in the Soviet Union. At power levels greater than about 2 kW, the SPT technology offers thrust levels that are superior to any western thruster in the specific impulse range of 10,000 to 20,000 m/s (1000 to 2000 s) as indicated in Fig. 2. The electrothermal arcjets operate at a higher thrust level for a given input power than the SPT-100, but at the expense of operating at a considerably lower specific impulse. Since propellant efficiency scales exponentially with specific impulse, the electrothermal arcjet is considerably less propellant efficient than the higher specific impulse SPT. The ion engine can easily operate at specific impulses much greater than 20,000 m/s (2000 s). However, the thrust produced for a given input power decreases with increasing specific impulse, so that the ion engine operating at more than 20,000 m/s produces less thrust than the SPT for the same power, although at an improved propellant efficiency. For near-Earth-space orbit transfer missions, this trade-off between propellant efficiency and power utilization results in an optimum specific impulse which is in the range of 10,000 to 20,000 m/s. Therefore, the unique capability of the SPT to operate efficiently and at high thrust levels in this specific impulse range will potentially offer substantial benefits for many near-Earth-space orbit transfer missions.

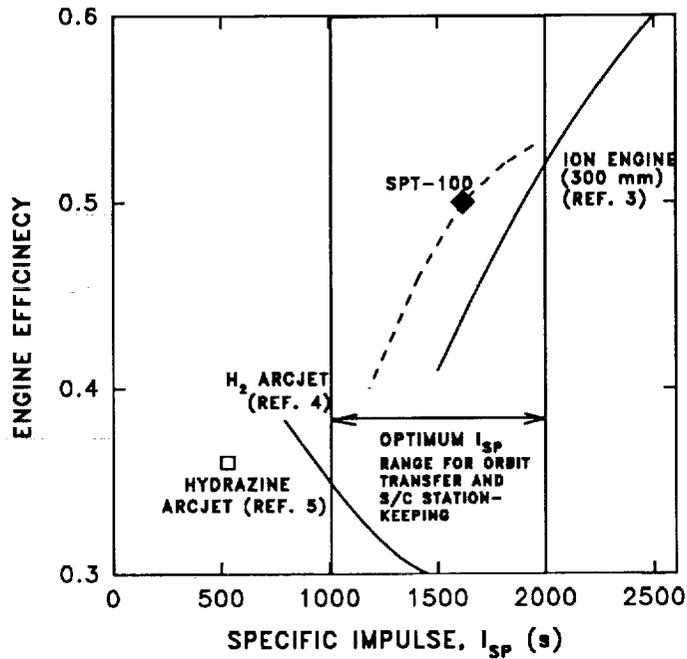


Figure 1 Comparison of electric thruster performance for thruster power levels of 0.5 to 2.0 kW (which is the appropriate range for satellite stationkeeping).

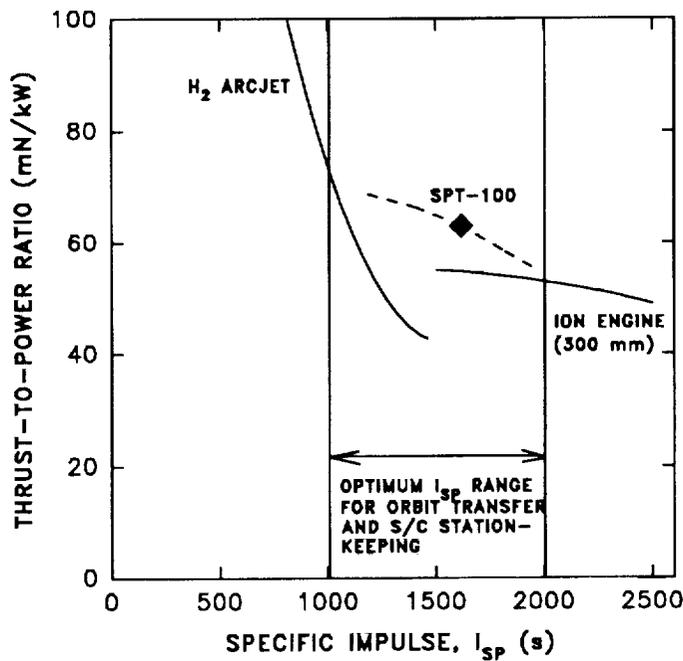


Figure 2 The stationary plasma thruster exhibits thrust-to-power performance superior to that of other electric propulsion engines in the specific impulse range 1000 to 2000 s.

2. Historical Background

In 1959, the Kurchatov Institute of Atomic Energy (IAE) in Moscow, USSR, was engaged in ion and plasma physics research associated with controlled fusion.¹ It was recognized at that time that electromagnetic acceleration of particles could be used to produce thrusters for space propulsion with specific impulses on the order of 10^3 to 10^4 seconds. In contrast, chemical thrusters were limited at the time to specific impulses of 300 to 400 seconds. Experimental electric propulsion research was initiated at IAE by G. L. Gradzovski.⁶ The first Soviet space flight of a plasma thruster took place five years later in 1964, with the flight of the Impulse Plasma Thruster (IPT). This year also marked the beginning of research on the Stationary Plasma Thruster (also referred to as a Coaxial Stationary Plasma Accelerator, a Hall thruster, or a Closed-Drift Hall thruster) in the USSR.⁶ Work on Hall Accelerators in the U.S. was initially performed at the NASA Lewis Research Center; these accelerators were first described in 1962.⁷ Patents on Hall thruster technology were granted to Gordon Cann of the United States in 1966, 1967, and 1968.⁸ By 1968, the Stationary Plasma Thruster (SPT) was developed at IAE to the point where the thruster performance was good enough for space applications. Much of the early work on closed-drift thrusters ceased about 1970 in the U.S. in order to concentrate on what was perceived to be the more promising, higher specific impulse gridded ion thruster.⁹ However, some work was continued at a low level in the U.S. through the 1980s and was performed primarily by Dr. Harold Kaufman.⁹

At the end of 1968, the SPT was slated by the Soviet Union to fly on the Meteor satellite. The thruster requirements for this flight experiment were a lifetime of 100 hours, an input of 400 watts, a thrust of 0.020 N (0.004 lb_f) with xenon propellant, and a system mass of 45 kg. An engineering model of the propulsion system was delivered in May 1969⁶ and included the SPT, power conditioning and control, and propellant storage and distribution. On March 1, 1972, the SPT was first operated in space. The thruster was operated for a total of 150 hours and demonstrated that the SPT could be integrated into and be compatible with the host spacecraft.

Since 1972, more than fifty SPT-70 thrusters have been flown in space. The "-70" designation indicates the characteristic diameter of the thruster in millimeters. It is claimed that there have been no failures of these thrusters in space. Only the SPT-50 and SPT-70 models have been flown while the SPT-100 model, which has a nominal input power of 1.35 kW, has been flight qualified in the USSR.¹⁰

3. Stationary Plasma Thruster Operation

A schematic diagram of the SPT is given in Fig. 3. During thruster operation, xenon gas is injected into both the hollow cathode and the main discharge chamber. The internal pressure in the hollow cathode is believed to be on the order of a few hundred pascals (a few torr). In the main discharge chamber, the propellant pressure is typically 0.1 Pa (10^{-3} torr). The cathode is heated by an external heater prior to initiation of the discharge. After the cathode is started, the heater is turned off and the cathode is operated in a self-heating mode.

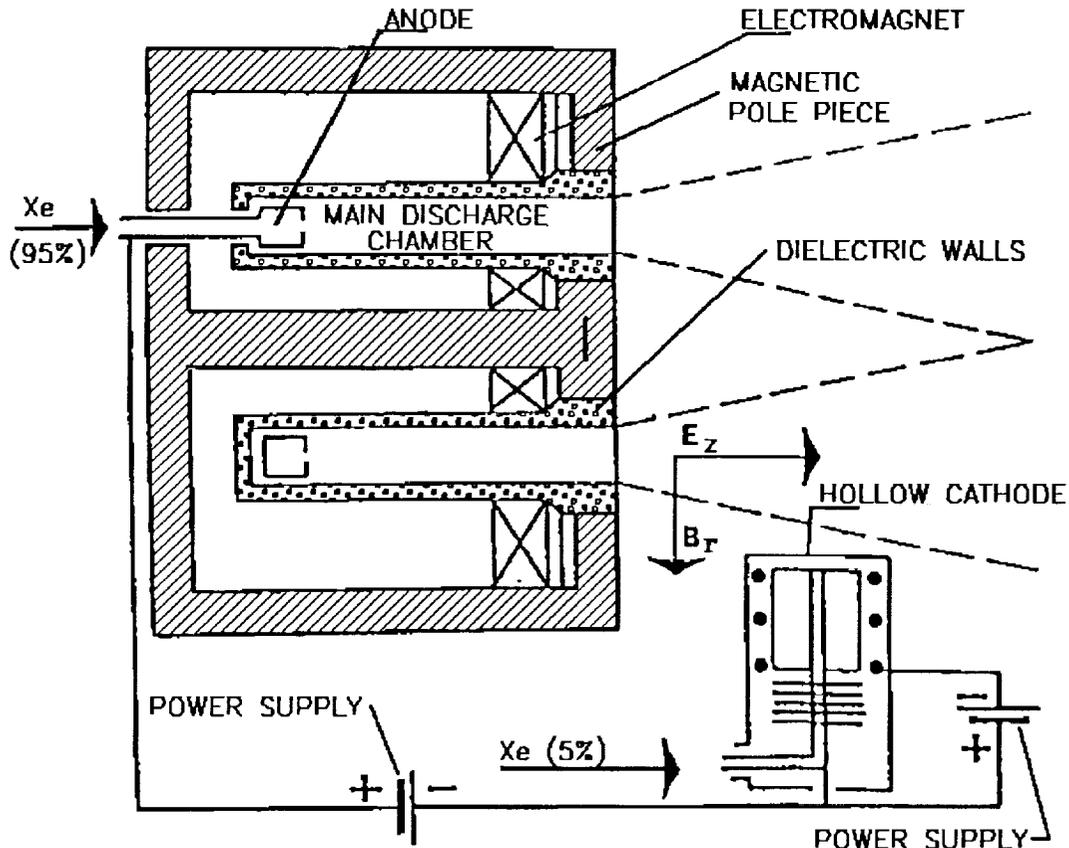


Figure 3 Stationary Plasma Thruster (SPT) schematic diagram.

A radial magnetic field is established across the annular channel of the main discharge region through the use of electromagnets. The electromagnet excitation is accomplished by flowing the discharge current through the magnets. No separate electromagnet power supply is used. A discharge voltage of typically 300 V is applied between the anode and the cathode. The radial magnetic field prevents electrons emitted by the cathode from streaming directly to the anode. Instead the electrons spiral around the magnetic field lines. The strength of the magnetic field is chosen such that the Larmor radius of the electron cyclotron motion is much smaller than the physical dimensions of the thruster. The electrons can cross the magnetic field lines and diffuse to the anode to sustain the discharge only by having collisions. The dominant electron scattering mechanism, however, is unclear. Classical electron diffusion across the magnetic field is clearly insufficient to account for the observed discharge currents. Plasma instability-enhanced, non-classical (Bohm¹¹) diffusion and a phenomenon called near-wall conductivity^{6,12} have been proposed to explain the operation of the SPT.

In any case, the magnetic field acts as an impedance to the flow of electrons from the cathode to the anode. This impedance serves to establish most of the applied potential difference between the anode and the cathode across the magnetic field (as opposed to in-the-sheath regions adjacent to each electrode). The result is an electric field in the plasma which is normal to the magnetic

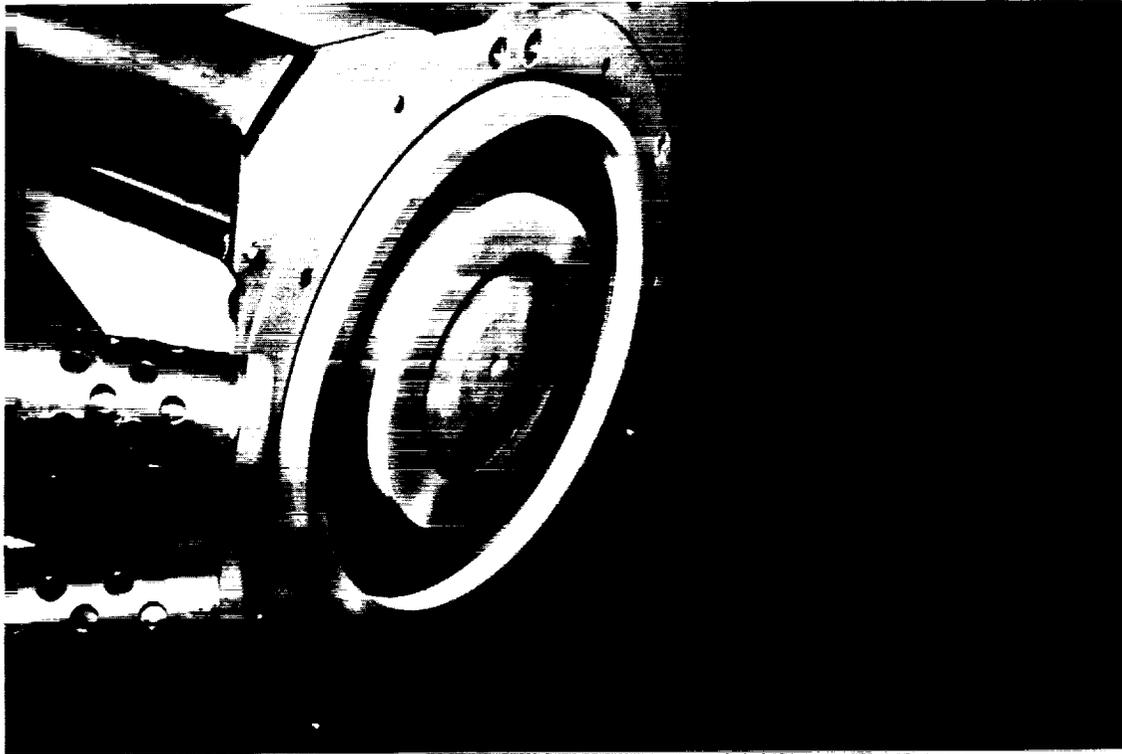


Figure 4 Photograph of the SPT-100 thruster.

field and is largely axial and points in the direction of the exhaust propagation. It is this electric field which serves to accelerate the ions created by electron bombardment of the propellant atoms to exhaust velocities on the order of 16,000 m/s. Additional electrons are drawn from the cathode by the resulting ion beam and provide current neutralization.

The electrons which diffuse from the cathode to the anode do so in a region of crossed electric and magnetic fields. This results in an electron drift motion in a direction mutually perpendicular to both the electric and magnetic fields. For the cylindrical geometry of the SPT, this is in the circumferential (or azimuthal) direction. This electron drift is analogous to the Hall effect and is why the thruster is sometimes referred to as a Hall thruster. For cylindrically symmetric geometries, the electron drift paths follow the circumferences of concentric circles within the annular discharge chamber region. These drift paths close back on themselves. This feature is essential to the proper operation of the device and is also the origin of the "closed-drift" designation for the thruster. The walls of the discharge chamber are typically made from a ceramic insulating material to prevent short-circuiting of the electric field by electron conduction through the chamber walls.

The accelerated ions are largely unaffected by the applied magnetic field by virtue of their relatively large mass. The requirement for the ion Larmor radius to be much larger than the physical dimensions of the device places an upper limit on the flux density of the applied magnetic field. However, some bending of the ion trajectories is unavoidable, and the result is a small net torque which will attempt to roll the thruster about its cylindrical axis. The net

azimuthal force which creates this torque may be as high as 8% of the axial thrust.¹⁰

During normal SPT operation approximately 90% of the input propellant flow rate is ionized and accelerated. As in a conventional gridded ion engine essentially all of the thrust is produced by the electrostatic acceleration of the ionized propellant atoms. This enables the thruster to produce exhaust velocities that are much higher than can be achieved with chemical thrusters or even electrothermal arcjets while maintaining relatively cool (typically less than 350°C) thruster component temperatures. The efficiency of the thruster depends, to a large extent, on the ratio of energy expended to produce the ions to that used to accelerate them. For a fixed ion production energy, reducing the specific impulse will result in decreased engine efficiency because a greater fraction of the input power goes into ionizing the propellant. This is true for both gridded ion engines and the SPT. To produce efficient engine operation at specific impulses less than 2,000 m/s (2000 s), it is necessary to efficiently ionize the propellant. The key to efficient ion production is to minimize ion recombination on the walls of the ion production chamber. In the SPT, most of the ions appear to be formed in a region of the discharge chamber in which a largely axial electric field exists. This electric field serves to accelerate the newly formed ions out of the discharge chamber and into the exhaust beam as soon as the ions are formed, thereby minimizing wall recombination. It is believed that this effect results in an ion production efficiency which is superior to that of conventional gridded ion engines because of the lower ion wall recombination rates and is largely responsible for the higher overall efficiency of the SPT at specific impulses less than 20,000 m/s (2000 s).

For electric thrusters, the performance and lifetime are intimately coupled with improved performance almost always obtained at the expense of decreased lifetime. For the SPT, proper tailoring of the magnetic field shape and strength is essential for simultaneously obtaining good performance and adequate lifetime.⁶ A lifetime of 3,000 to 4,000 hours is claimed for the SPT.⁶

4. Schedule of Activities in the Soviet Union

To evaluate the performance of the SPT-100 at the Scientific-Research Institute of Thermal Processes and at "Fakel" Enterprise, the U.S. instrumentation listed in Table 2 was brought to Russia. An outline of the schedule of experimental activities at these two laboratories is given in Table 3. The thruster operating modes which were tested are identified in Appendix A. The instrumentation listed in Table 2 was selected to provide as much insight into the performance of the SPT as possible. To determine the thruster's performance, it is necessary to measure only three parameters: the input power, the propellant mass flow rate, and the thrust. Hand-held multimeters and a digital oscilloscope were brought to determine the input power, and a volumetric flow meter calibration kit was brought to calibrate the xenon mass flow meters. It was not feasible to bring a thrust stand, so thrust measurements were taken with the thrust stand at "Fakel" Enterprise; there was no thrust stand in operation at NIITP. A set of precision weights was brought to be used in the calibration of the "Fakel" thrust stand. In addition, a U.S. ion gauge tube and controller were used to measure the vacuum chamber pressure during thruster performance measurements, and a residual gas analyzer (RGA) was included to determine the identity of the propellant gas and to check for any contaminant gases in the vacuum system.

Table 2. Equipment List Description

1. Residual Gas Analyzer (RGA): Measures partial pressures of gases remaining in a vacuum chamber. Used to measure the water vapor and oxygen contamination levels in the Soviet vacuum facility, as well as to confirm the identity of the propellant.
 2. Ion Gauge Tube and Controller: Measures total vacuum chamber pressures. Used to measure the pressures at NIITP and "Fakel" during thruster testing. The total pressure is more easily and reliably determined with this instrumentation than by summing the partial pressures obtained with the residual gas analyzer.
 3. Flow Meter Calibration Kit: Used to measure gas flow rates in the range of 5 to 100 standard cubic centimeters per minute, which is the range of interest for the SPT-100. Used to verify the calibration of the "Fakel" gas metering equipment.
 4. Digital Storage Oscilloscope and Probes: Measures voltages as a function of time. Used to measure the time varying components of the SPT's electrical characteristics during operation.
 5. Digital Multimeters: Measures currents, voltages, and resistances. Used to verify measurements of electrical parameters.
 6. Pressure Gauge (barometer): Measures atmospheric pressure. Used to measure the local atmospheric pressure as required by the Flow Meter Calibration Kit.
 7. Precision Weights: Used in the calibration of the "Fakel" thrust stand.
 8. 286 Notebook Computer and Software: Used to perform analyses of the SPT-100 test data.
 9. Current Shunt (10 A): Used for current measurements during operation of the SPT-100.
-

Table 3. Outline of Activities in the USSR

Date	Day	Activity
09/30	Monday	Kick-off meeting at NIITP. Discussion of test procedures. Installation of U.S. laboratory equipment.
10/01	Tuesday	First operation of the SPT-100. Thruster testing in modes 1 and 2 on xenon. Digital oscilloscope traces, current, voltage, and pressure measurements with U.S. instruments. RGA measurements of residual background gases. Collection of the remaining operating data with NIITP instrumentation.
10/02	Wednesday	SPT-100 operation on xenon in modes 1, 2, and 3. Tour of the MPD lab at NIITP.
10/03	Thursday	SPT-100 operation on krypton and argon. Presentation by A.I. Morozov on the history of the SPT.
10/04	Friday	Tour of the Moscow Aviation Institute. Packed up equipment at NIITP. Presentation on the "Bootstrap" method for SPT life testing.
10/07	Monday	Kick-off meeting with "Fakel" representatives. Discussion of test program.
10/08	Tuesday	Installation of U.S. instruments at "Fakel." Thrust stand calibration using U.S. weights. Flow calibration with U.S. calibration equipment. Tour of new "Fakel" electric propulsion facility under construction.
10/09	Wednesday	Operation of two SPT-100 thrusters on xenon in modes 1 and 2. Check of electromagnetic effects on thrust stand operation. Determination of thermal effects on thrust stand operation.
10/10	Thursday	Operation of SPT-100 on xenon in mode 3. Check for plasma interactions with thrust stand operation.
10/11	Friday	Operation of SPT-100 on krypton in mode 1. Recalibration of flow meter on xenon with U.S. calibration equipment. Packed up instrumentation.
10/14	Monday	Review of results with SDIO and JPL managers.
10/15	Tuesday	Discussion of test results at NIITP.

5. Measurements at the Scientific-Research Institute of Thermal Processes (NIITP)

At NIITP, U.S. instrumentation was used to measure the discharge current, discharge voltage, tank pressure, residual vacuum chamber gases, and the discharge current and voltage oscillations. Data were taken at 36 operating points on xenon, 14 on krypton, and 1 on argon. For thruster operation on krypton or argon, xenon was still used to operate the cathode. The measurements of current, voltage, and tank pressure agreed well with those obtained with NIITP's instrumentation. A photograph of the vacuum test facility at NIITP is given in Fig. 5.

Initial oscilloscope measurements were made at a location in the power circuit which was between the discharge power supply and an electrical filter. At this location, the discharge current and voltage oscillations were observed to be only a few percent of their dc values. Subsequent measurements were made with the oscilloscope probes positioned downstream of the filter (i.e., between the filter and the thruster). Large scale oscillations in the discharge current were observed at this location for almost all operating modes examined, including the nominal operating point where the advertised performance is determined. An example of these large scale oscillations is given in Fig. 6. In this figure, the top trace is the discharge voltage waveform at a scale setting of 20 V/div and a dc voltage of 325 V. The bottom trace is the ac component of the discharge current with a scale setting equivalent to 1.25 A/div. The measured dc discharge current in this case was 2.5 A. The oscillations in the discharge current are essentially 100% of the dc level. The frequency of these oscillations was found to be in the range 20 to 30 kHz for most operating modes. No thrust data were taken at NIITP since the



Figure 5 Photograph of the vacuum test facility at the Scientific-Research Institute of Thermal Processes in Moscow.

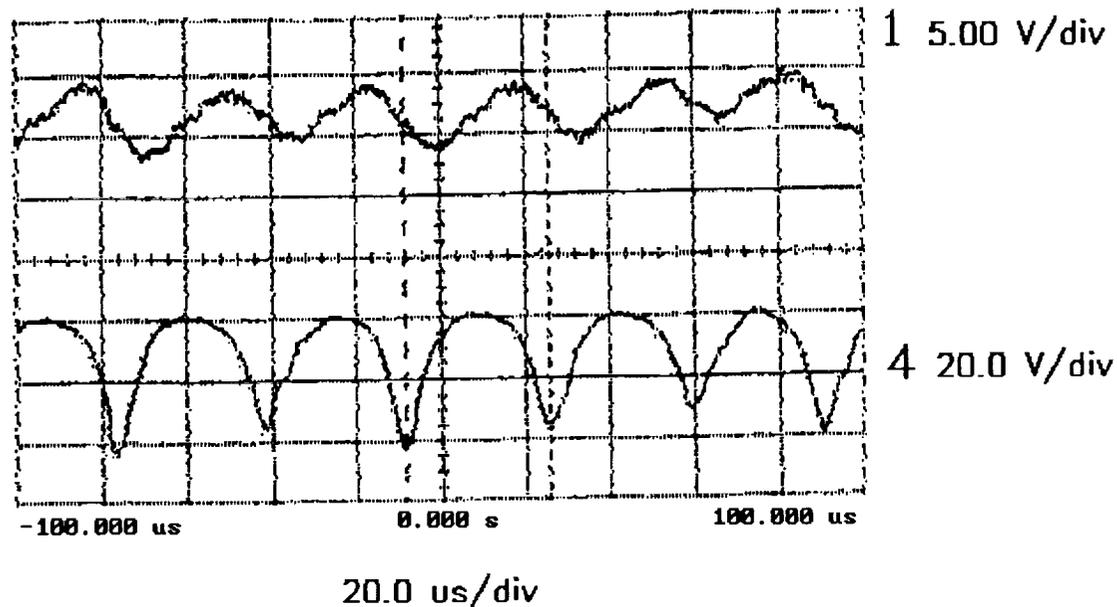


Figure 6 Typical discharge voltage (top: 20 V/div) and current (bottom: 1.25 A/div) waveforms showing the 20- to 30-kHz oscillations observed at most operating modes at NIITP.

thruster was not mounted on a thrust stand at this laboratory. Calibrations of the propellant flow meters at NIITP were also not performed because the NIITP flow meters are designed to function properly only when exhausting into a near vacuum and the calibration kit requires that the flow meters exhaust to atmospheric pressure.

6. Measurements at "Fakel" Enterprise

Thruster testing at "Fakel" Enterprise in Kaliningrad was performed off-site at the "Yantar" facility. A photograph of the vacuum test facility at "Fakel" is shown in Fig. 7. The same measurements were made here as were made at NIITP, with the following additions. The propellant mass flow meter, which measures the total flow rate into the thruster, was calibrated with the "bubble volume" calibration kit before and after the thruster performance tests. The thrust stand was calibrated using the U.S. precision weights, and thermal, magnetic, and plasma effects on the thrust stand were investigated.

The thrust stand is a torsional design which has been developed at "Fakel" over the past twenty years. A laser, positioned outside of the vacuum system, is used to detect the rotation of the thrust stand by bouncing the laser beam off of a mirror attached to the torsion rod and detecting the return beam with an array of photodiodes, also positioned outside of the vacuum system. Positioning this sensing equipment outside of the vacuum system eliminates the possibility of thruster-induced plasma effects from influencing the indicated thrust. The signal from the photodiode array is used in a closed-loop control system to drive an electromagnet to return the

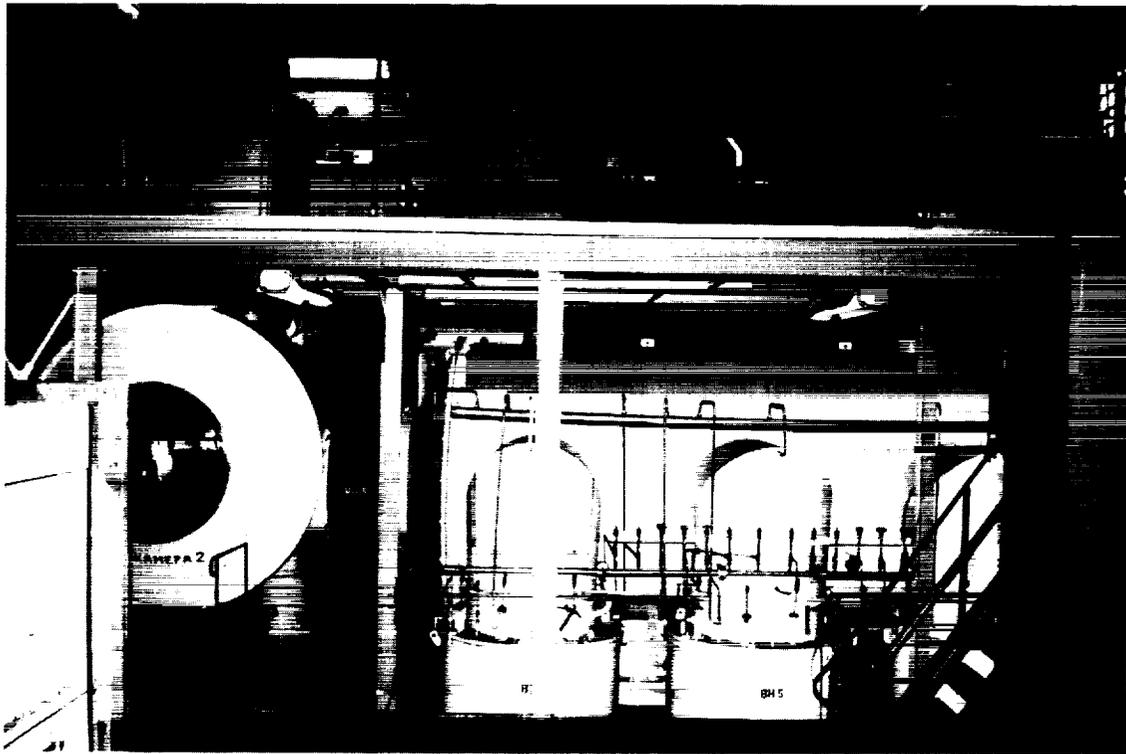


Figure 7 Photograph of the vacuum test facility at "Fakel" Enterprise.

thrust stand to its zero position. The current through the electromagnet is calibrated to indicate the thrust from the thruster. This calibration is performed at atmospheric pressure through the application of dead weights. No provision is made for performing the dead weight calibration under vacuum. The thrust stand calibration performed using U.S. weights is given in Fig. 8.

Magnetic effects on the thrust stand were investigated by applying different currents to the electromagnets on the thruster with the thruster discharge off. Currents of 2, 3, 4 and 4.5 A were applied with no detectable change in the thrust stand zero position.

The results of the flow meter calibration are given in Fig. 9. The U.S. calibrations performed before and after the tests were consistent to within approximately 2% of the readings. The U.S. calibrations, however, differed from those obtained previously by "Fakel," using a similar calibration technique, by approximately 5% of the reading. The reason for this difference is not known. The use of the U.S. calibration instead of "Fakel"'s results in slightly better indicated performance for the thruster.

The experimental setup at "Fakel" physically did not permit the oscilloscope to be placed between the electrical filter in the power circuit and the thruster, as was done at NIITP. The oscilloscope traces obtained between the filter and the discharge power supply indicated very low level (a few percent) current and voltage oscillations just as those at NIITP. It is not known if large scale oscillations would have been observed if the oscilloscope probes could have been positioned between the filter and the thruster.

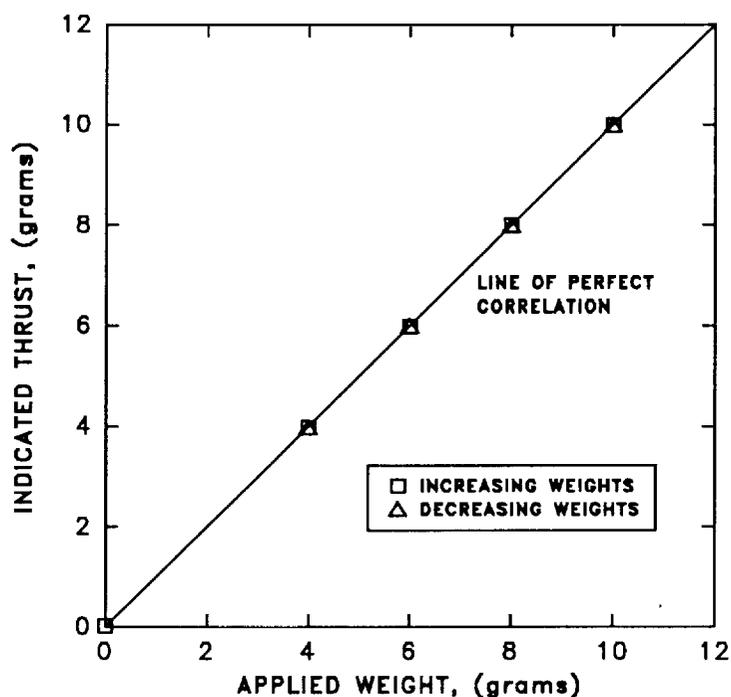


Figure 8 Calibration of the thrust stand at "Fakel" using U.S. weights indicates excellent agreement with the applied weights and no measurable hysteresis.

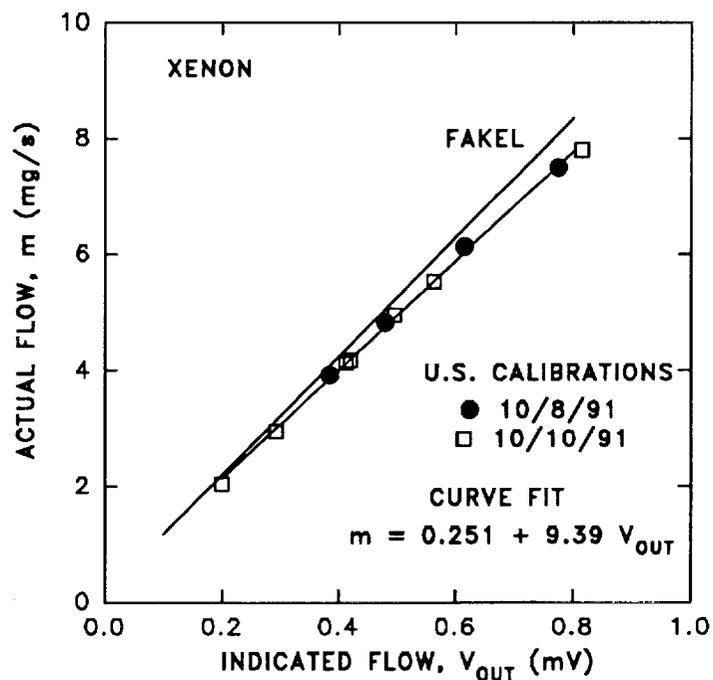


Figure 9 Calibration data for the thermal mass flow meter at "Fakel" Enterprise showing the difference between the U.S. and "Fakel" calibrations.

7. Results and Discussion

The results of the data collected at "Fakel" for operation of two SPT-100 thrusters on xenon are given in Table 4. These data indicate the degree of agreement between the current and voltage measurements made by U.S. and "Fakel" instruments. The cathode-to-ground voltage was measured with "Fakel" instruments only and indicates the floating potential of the cathode with respect to the vacuum chamber walls. The effective pumping speed of the "Fakel" vacuum system is given in the seventh column and is based on the U.S. calibrated measured propellant flow rate and the corresponding vacuum chamber pressure as measured with the U.S. ion gauge and corrected for xenon. The chamber pressures during performance testing ranged from 3.1×10^{-5} to 5.6×10^{-5} torr. These pressures are considered to be at the upper end of the pressure range at which meaningful performance data can be obtained. At 5×10^{-5} torr, it is estimated that propellant backflow from the facility adds approximately 2.5% to the total propellant flow rate; at 3×10^{-5} torr, it adds 1.5% (see Appendix B). The thruster performance given in Table 4 has been corrected for this effect.

At the nominal 1.35 kW operating point, the thruster was operated continuously for 2 hours. After this time, a thermally induced zero shift in the thrust stand of approximately 0.003 N (0.3 grams force) was observed. This value for the thrust stand thermal effects was used to approximately correct the thrust stand readings for all other operating points. The "Corrected" thrust indicated in Table 4 was obtained by subtracting 0.003 N from the indicated thrust.

Two identical SPT-100 thrusters were tested in order to obtain some information on the thruster-to-thruster performance variation. The two thrusters are designated "1" and "2" in Table 4. In the middle of Table 4 are two columns indicating the voltage and auxiliary current through the electromagnet. At the main operating point (mode 1 in Appendix A), the discharge current alone is used to power the electromagnet. At the off-normal modes (modes 2 and 3 in the appendix), an additional electromagnet is used to fine tune the magnetic field strength in order to improve the discharge stability at these off-normal points. The power supplied by this auxiliary magnet supply is included in the overall efficiency determination.

The data from Table 4 are plotted in Figs. 10-13. The thruster performance as a function of specific impulse is plotted in Fig. 10. For clarity, the uncertainty in the specific impulse is not shown. Clearly the performance as determined in this activity supports the advertised performance claims. The uncertainty in efficiency shown in Fig. 10 is calculated according to the procedure given in Appendix B. In Fig. 11, the specific impulse is given as a function of input power with the discharge current as a parameter. For a fixed discharge current, the thruster input power is varied by changing the applied discharge voltage. The discharge power supply is operated in a voltage regulating mode, and the propellant flow rate is adjusted to maintain the desired discharge current. For a fixed discharge current, the mass flow rate is approximately constant, independent of the discharge voltage. The data in Fig. 12 indicate the variation of specific impulse with discharge voltage. These data indicate the expected trend of increasing specific impulse with discharge voltage. They also indicate that the specific impulse increases with increasing discharge current (and flow rate) at a fixed discharge voltage. This is an indication that the propellant efficiency increases as the discharge chamber plasma density is increased, as one might expect.

Table 4 Stationary Plasma Thruster Evaluation: SPT-100 (Xenon)

Discharge USA (V)	Discharge Fakel (V)	Discharge USA (A)	Discharge Fakel (A)	Cathode-to- Ground (V)	Pressure USA (torr)	Pumping Speed (l/s)	Electromagnet Voltage (V)	Electromagnet Aux. Cur. (A)	Input Power (W)	Thrust		Flow Rate		Specific Impulse (s)	Engine Efficiency	Thruster Number
										Indicated (N)	Corrected (N)	Indicated (mg/s)	Fakel (mg/s)			
300	300	4.48	4.50	-18.1	5.6E-05	11918			1344	0.087	0.084	5.13	5.47	1671	0.51	1
302	300	4.46	4.50	-15.2	4.1E-05	16427			1347	0.089	0.086	5.18	5.52	1695	0.53	
200	200	5.94	6.00	-18.0	4.5E-05	19443	4.40		1188	0.089	0.086	6.73	7.21	1305	0.46	
300	300	4.48	4.48	-15.8	3.4E-05	19988	3.40		1344	0.087	0.084	5.23	5.57	1641	0.50	2
202	200	5.98	6.03	-20.0	4.9E-05	17855	5.05		1208	0.087	0.084	6.90	7.21	1275	0.43	
203	203	2.45	2.50	-21.4	3.1E-05	13260	2.75	0.98	500	0.033	0.030	3.16	3.32	970	0.29	
400	400	2.43	2.50	-22.7	3.1E-05	13063	3.00	0.99	975	0.055	0.052	3.05	3.27	1705	0.45	
202	203	2.48	2.50	-23.0	3.2E-05	12464	2.60	0.99	504	0.034	0.031	3.00	3.07	1033	0.31	1
401	400	2.46	2.50	-18.3	3.1E-05	12078	2.85	1.01	989	0.056	0.053	0.280	2.88	3.01	1880	0.49
300	300	4.48	4.53	-17.8	4.2E-05	16036	3.30	0.00	1344	0.087	0.084	5.18	5.52	1656	0.51	
250	250	4.48	4.53	-17.1	4.2E-05	16181	3.35	0.00	896	0.077	0.074	5.23	5.57	1446	0.47	
200	200	4.48	4.53	-18.5	4.2E-05	15890	3.90	0.52	1575	0.096	0.093	5.13	5.47	1850	0.54	
351	350	4.48	4.53	-18.6	4.2E-05	15890	4.00	0.49	1789	0.103	0.100	5.13	5.47	1989	0.54	
399	398	4.48	4.53	-18.7	3.8E-05	15957	3.65	0.51	1184	0.077	0.074	4.66	4.95	1620	0.50	
300	300	3.94	3.98	-18.0	3.8E-05	15957	3.20	0.00	994	0.068	0.065	4.70	4.95	1423	0.46	
251	250	3.96	4.00	-17.2	4.2E-05	14437	3.20	0.00	800	0.057	0.054	4.66	4.95	1183	0.39	
202	200	3.96	4.00	-18.9	3.8E-05	15636	3.70	4.95	1408	0.084	0.081	4.60	4.85	1810	0.51	
351	348	3.96	4.00	-21.8	3.8E-05	15475	3.75	0.49	1586	0.091	0.088	4.52	4.80	1986	0.54	
400	398	3.96	4.00	-21.3	3.1E-05	12669	3.00	0.99	741	0.048	0.045	3.02	3.16	1522	0.45	
300	300	2.46	2.50	-20.7	3.2E-05	12082	3.00	0.99	620	0.042	0.039	2.90	2.97	1340	0.41	
253	253	2.44	2.48	-22.4	3.2E-05	12078	3.00	1.02	498	0.033	0.030	2.90	3.11	1031	0.31	
201	200	2.46	2.48	-17.4	3.1E-05	12078	3.00	0.99	864	0.051	0.048	2.88	3.01	1703	0.46	
350	350	2.46	2.50	-22.2	3.1E-05	12078	3.00	1.02	992	0.056	0.053	2.88	3.01	1880	0.49	
402	400	2.46	2.50	-17.0	4.5E-05	16323	4.10	0.00	1504	0.098	0.095	5.75	6.03	1717	0.53	
302	300	4.98	5.03	-16.5	4.5E-05	16323	4.15	0.00	1240	0.085	0.082	5.65	6.03	1482	0.48	
249	248	4.98	5.03	-17.6	4.5E-05	17344	4.20	0.00	1021	0.074	0.071	5.65	6.03	1283	0.44	
205	205	4.98	5.03	-17.8	4.2E-05	17344	4.30	0.00	1738	0.104	0.101	5.60	5.98	1840	0.52	
349	348	4.98	5.03	-16.5	4.2E-05	17198	4.90	0.49	2007	0.115	0.112	5.65	5.93	2058	0.56	
401	400	5.00	5.03	-16.2	4.9E-05	17482	5.40	0.00	1788	0.117	0.114	6.59	7.05	1766	0.55	
300	300	5.96	6.00	-15.7	4.9E-05	17482	5.40	0.00	1485	0.103	0.100	6.75	7.05	1550	0.51	
250	250	5.94	5.98	-16.9	4.9E-05	17606	5.50	0.00	1200	0.090	0.087	6.64	7.10	1339	0.48	
202	205	5.94	5.98	-16.9	4.9E-05	17482	5.50	0.00	2079	0.126	0.123	6.75	7.05	1906	0.55	
350	348	5.94	6.00													

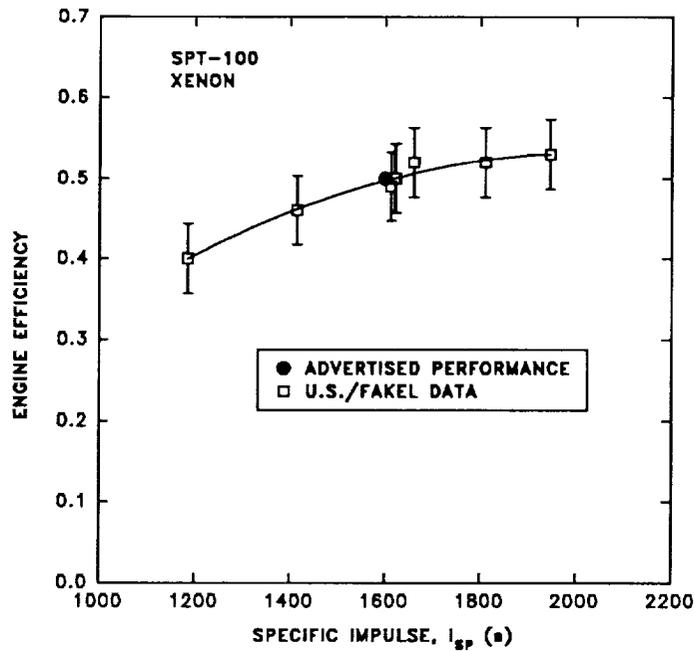


Figure 10 SPT-100 thruster performance showing good agreement between the advertised performance and that obtained by the U.S. technical specialists at "Fakel" (at 4.48 A).

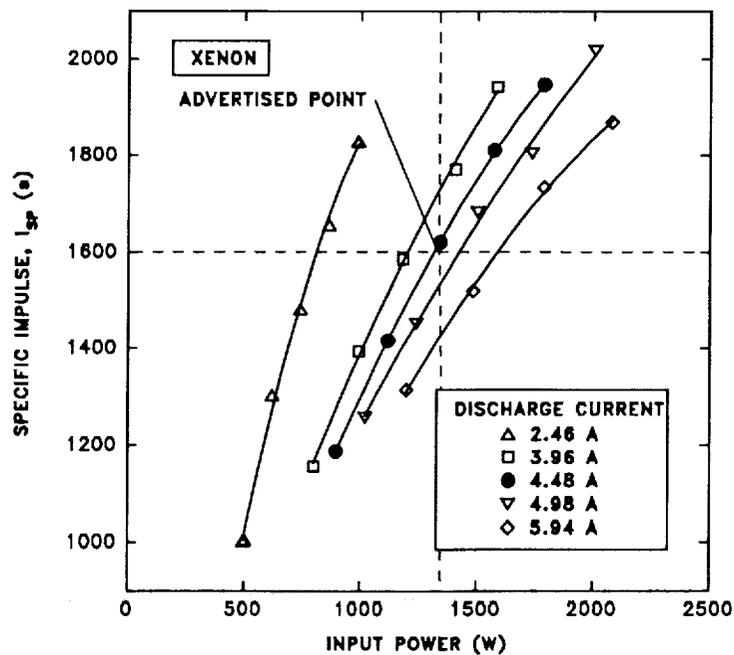


Figure 11 For a fixed discharge current (i.e., propellant flow rate), the input power can be varied over approximately a factor of two by adjusting the applied discharge voltage.

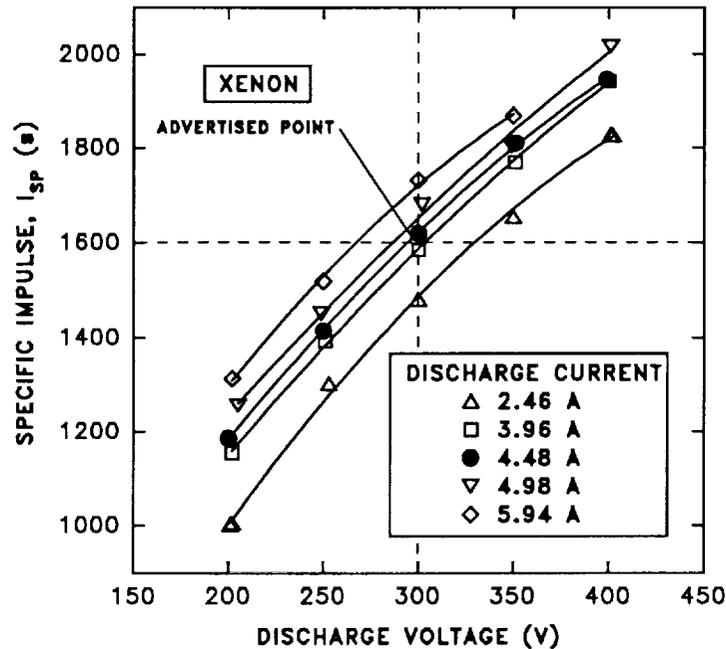


Figure 12 The specific impulse is a physically realistic non-linear function of the applied discharge voltage.

Finally, the thruster efficiency as a function of specific impulse is given in Fig. 13. These data indicate that the highest efficiencies are achieved at the highest specific impulses, and that at a given specific impulse, the highest efficiency is obtained at the highest discharge current (power level) for that specific impulse. This again is a reflection that the propellant efficiency increases with discharge current and flow rate at a given discharge voltage. All of these trends are consistent with our present understanding of the physics of SPT operation.

8. Conclusions

Based on the information obtained in the Soviet Union in the course of the phase I activity, the following conclusions can be drawn:

1. No differences in definitions of thruster efficiency were identified and all of the factors which contribute to the determination of the overall efficiency appear to be correctly taken into account.
2. The actual performance of the SPT-100 appears to be close to the claimed performance at the nominal operating point.
3. Uncertainties in the performance are largely due to uncertainties in the thrust measurement. The inability to perform a dead weight calibration in vacuum of the thrust stand at "Fakel" is a source of additional, but intangible uncertainty. This uncertainty, however, is not expected to materially change these conclusions.

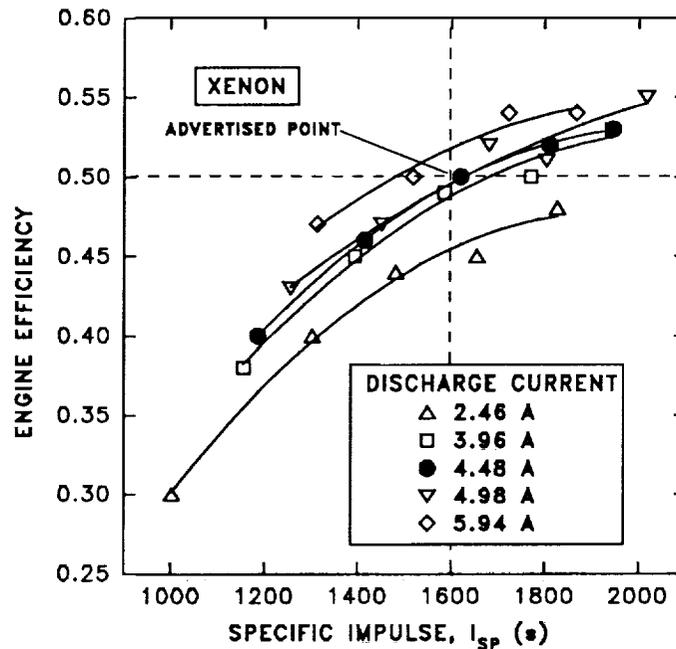


Figure 13 The SPT-100 efficiency is a physically realistic non-linear function of the I_{sp} indicating that the efficiency does not increase without bounds as the I_{sp} is increased.

9. Recommendations

Based on the results from the phase I testing, the phase II activity to bring SPT thrusters to the U.S. for further performance characterization and endurance testing is definitely worth pursuing. Significant technical and integration issues for the thruster exist which were not addressed in the phase I program. These issues include quantification of thruster erosion rates, measurement of the performance variation as a function of long-duration operation, quantification of the reported¹⁰ large exhaust beam divergence angle, determination of the non-propellant efflux, and measurement of the thrust vector stability and thruster-induced roll torques. It is also of interest to determine if the discharge current instability observed for most operating modes at NIITP is inherent in SPT operation or is merely an artifact of the power processing hardware used at NIITP. These issues require quantification in order to maximize the probability of user application of the stationary plasma thruster technology.

The most critical of these outstanding technical issues is the long-duration behavior of the thruster. The low-thrust nature of electric propulsion requires that the thrusters be capable of operating for several thousand hours to accomplish typical missions of interest (such as north/south stationkeeping or near-Earth-space orbit transfer missions). Furthermore, as is typical of most mechanical systems, the performance and lifetime of the thruster are intimately coupled. Good performance of an electric thruster can often be achieved at the expense of lifetime. It is, therefore, essential that the lifetime and performance degradation with time be quantified for operation at the optimized performance point.

Some long-duration tests of the SPT-70 and SPT-100 thrusters have been performed by "Fakel" Enterprise and have reportedly demonstrated a lifetime of 3000 to 4000 hours for the SPT-70. However, there is some question as to the applicability of these endurance tests if they were performed in oil-diffusion-pumped vacuum facilities. In such facilities, an oil film deposit slowly forms on the insulator surfaces of the main discharge chamber. This film must be mechanically removed by hand-sanding the insulator surface every one to two hundred hours. This frequent intervention makes interpretation of the measured erosion rates and performance degradation with time difficult at best. These problems can be avoided by performing long-duration endurance testing in very clean, cryopumped vacuum facilities.

The phase II activity will perform long-duration testing of a stationary plasma thruster in a cryopumped facility with sufficient pumping speed to maintain vacuum chamber pressures approximately an order of magnitude lower than those used in the phase I performance tests. Furthermore, flight application of stationary plasma thruster technology will benefit greatly from an examination of wear-out phenomena and erosion rates at off-nominal operation modes and as a function of propellant choice. Specific mission applications may substantially profit from having the knowledge available to trade thrust level, specific impulse, and lifetime to obtain the greatest mission benefits.

The ultimate objective, of course, is to use the stationary plasma thruster technology to significantly increase the propulsion capabilities of U.S. spacecraft. Electric propulsion systems consist of a power source (typically photovoltaic solar arrays for power levels up to a few tens of kilowatts), a power distribution system, power conditioning hardware, the thrusters, and a propellant storage and distribution system. The flight power conditioning hardware built in the Soviet Union for the SPT-70 thruster is excessively massive by western electronics standards. The thrusters themselves constitute a relatively small fraction of the total electric propulsion system mass and cost (on the order of 5% for both). Even if the SPT-70 or SPT-100 thrusters were used directly on a U.S. spacecraft, the majority of the propulsion subsystems, as listed above, would be supplied by U.S. industries.

10. References

1. Barnett, J. W., "A Review of Soviet Plasma Engine Development," AIAA-90-2600, July 1990.
2. Patterson, M. J., Robinson, R. S., Schemmel, T. D., and Burgess, D. R., "Experimental Investigation of a Closed-Drift Thruster," AIAA-85-2060, 1985.
3. Patterson, M. J., and Foster, J. E., "Performance and Optimization of a 'Derated' Ion Thruster for Auxiliary Propulsion," AIAA-91-2350, June 1991.
4. Curran, F. M., Bullock, S. R., Haag, T. W., Sarmiento, C. J., and Sankovic, J. M., "Medium Power Hydrogen Arcjet Operation," AIAA-91-2227, June 1991.
5. Smith, R. D., et al., "Development and Demonstration of a 1.8 kW Hydrazine Arcjet Thruster," AIAA-90-2547, July 1990.

6. Morozov, A. I., Kurchatov Institute of Atomic Energy, personal communication, October 1991.
7. Seikel, G. R., and Reshotko, E., "Hall Current Ion Accelerator," *Bulletin of the American Physical Society*, Ser. II, Vol. 7, June 1962, p. 414.
8. U.S. patent number 3243954, "Plasma Accelerator Using Hall Currents," Gordon L. Cann, assignor to Electro-Optical Systems, Inc., Pasadena, CA, patented April 5, 1966, filed August 17, 1962, Ser. No. 217631. U.S. patent number 3309873, "Plasma Accelerator Using Hall Currents," Gordon L. Cann, assignor to Electro-Optical Systems, Inc., Pasadena, CA, patented March 21, 1967, filed August 31, 1964, Ser. No. 394,383. U.S. patent 3388291, "Annular Magnetic Hall Current Accelerator," Gordon L. Cann, assignor to Electro-Optical Systems, Inc., Pasadena, CA, patented June 11, 1968, filed August 31, 1964, Ser. No. 393,288.
9. Kaufman, H. R., "Technology of Closed-Drift Thrusters," *AIAA Journal*, Vol. 23, No. 1, January 1985, p. 78.
10. V. Kim, Moscow Aviation Institute, personal communication, October 1991.
11. Bohm, D., "Minimum Ionic Kinetic Energy for a Stable Sheath," in *The Characteristics of Electrical Discharges in Magnetic Fields*, edited by A. Guthrie and R. K. Wakerling (McGraw-Hill, New York, 1949) pp. 77-86.
12. Morozov, A. I., et al., "Plasma Accelerator with Closed Electron Drift and Extended Acceleration Zone," *Soviet Physics -- Technical Physics*, Vol. 17, No. 1, July 1972.

Appendix A
SPT-100 Test Matrix

Input Current (A)	Input Voltage (V)				
	200	250	300	350	400
2.5	2	3	3	3	2
4.0	3	3	3	3	3
4.5	3	3	1	3	3
5.0	3	3	3	3	3
6.0	2	3	3	3	-----

- 1 -- Main operating mode
2,3 -- Off-normal operating modes

Appendix B Performance Uncertainty Analysis

The total thruster efficiency is determined from experimentally measured quantities according to the equation

$$\eta_t = \frac{T^2}{2\dot{m}P}$$

where T is the measured thrust in newtons, m is the total propellant mass flow rate into the device in kg/s (including backflow from the vacuum chamber), and P is the total engine input power in watts (including the power to the electromagnets).

In general, the uncertainty of a quantity y, which is a function of measurable independent variables x_1, x_2, \dots, x_n , may be computed from

$$\Delta y = \left[\left(\frac{\partial y}{\partial x_1} \Delta x_1 \right)^2 + \left(\frac{\partial y}{\partial x_2} \Delta x_2 \right)^2 \dots + \left(\frac{\partial y}{\partial x_n} \Delta x_n \right)^2 \right]^{1/2}$$

For the thruster efficiency, this equation becomes

$$\Delta \eta_t = \left[\left(\frac{\partial \eta_t}{\partial T} \Delta T \right)^2 + \left(\frac{\partial \eta_t}{\partial \dot{m}} \Delta \dot{m} \right)^2 + \left(\frac{\partial \eta_t}{\partial P} \Delta P \right)^2 \right]^{1/2}$$

which may be simplified as

$$\frac{\Delta \eta_t}{\eta_t} = \left[4 \left(\frac{\Delta T}{T} \right)^2 + \left(\frac{\Delta \dot{m}}{\dot{m}} \right)^2 + \left(\frac{\Delta P}{P} \right)^2 \right]^{1/2}$$

The terms on the right-hand side of this equation represent, in order, the measurement uncertainties in the thrust, propellant mass flow rate, and the thruster input power. The Soviets claim a thrust measurement uncertainty of between 2 and 4%. Since this indicates an uncertainty in the determination of the thrust measurement uncertainty, the larger value will be used, i.e.,

$$\frac{\Delta T}{T} = 0.04$$

There was no evidence to suggest that the uncertainty of the thrust stand measurement is greater than 4%.

The propellant mass flow rate is the sum of the propellant injected in the thruster and that which backflows from the vacuum facility by virtue of operating with a non-zero background pressure. Therefore, the flow rate measurement uncertainty is given by

$$\frac{\Delta \dot{m}}{\dot{m}} = \frac{1}{\dot{m}} \left[\dot{m}_i^2 \left(\frac{\Delta \dot{m}_i}{\dot{m}_i} \right)^2 + \dot{m}_b^2 \left(\frac{\Delta \dot{m}_b}{\dot{m}_b} \right)^2 \right]^{1/2}$$

where m_i is the injected propellant mass flow rate, and m_b is the mass flow rate due to backflow from the facility. The uncertainty in the injected propellant mass flow rate measurement from U.S. calibration data is estimated to be 2%, i.e.,

$$\frac{\Delta \dot{m}_i}{\dot{m}_i} = 0.02$$

The mass flow rate due to backflow from the facility is a function of facility pressure and is given for xenon in Table 5 for the SPT-100 thruster assuming a neutral atom temperature of 300 K.

Table 5 Backflow as a Function of Pressure

P_i (torr)	m_b (mg/s)
10^{-6}	0.003
10^{-5}	0.027
2×10^{-5}	0.055
5×10^{-5}	0.136
10^{-4}	0.273
2×10^{-4}	0.545
3×10^{-4}	0.818

The values of the backflow propellant mass flow rates are calculated based on free molecular flow and consequently have a rather high uncertainty. Therefore, an uncertainty of 100% is assumed for the propellant backflow, i.e.,

$$\frac{\Delta \dot{m}_b}{\dot{m}_b} = 1.00$$

The U.S. data indicate a minimum tank pressure of 3.4×10^{-5} torr during thruster performance testing at the nominal operating point. Thus, for an injected propellant flow rate of 5.2 mg/s, the uncertainty in the flow rate determination is

$$\frac{\Delta \dot{m}}{\dot{m}} = 0.027$$

Finally, the uncertainty in the input power is taken to be 1%. Therefore, the overall thruster efficiency uncertainty becomes

$$\Delta \eta_t = [4(0.04)^2 + (0.027)^2 + (0.01)^2]^{1/2}$$

or

$$\frac{\Delta \eta_t}{\eta_t} = 0.085$$

For a nominal efficiency of 0.50 (which corresponds to a specific impulse of 1640 s, a thrust of 0.084 N, and an input power of 1344 W) the uncertainty in the efficiency is

$$\Delta \eta_t = 0.043$$

Therefore, with this uncertainty, the total thruster efficiency can at best be specified within the following bounds:

$$0.46 \leq \eta_t \leq 0.54$$

Appendix C Glossary of Key Technical Terms

anode

Positive electrode at which electrons are collected.

Bohm diffusion

Non-classical diffusion of electrons across a magnetic field in which the diffusion rate is inversely proportional to the magnetic flux density.

cathode

Negative electrode at which electrons are emitted.

electric thruster

A rocket engine which uses electrical power to produce thrust with propellant exhaust velocities many times higher than that possible with chemical rockets.

electron cyclotron motion

The circular motion of an electron about a magnetic field line of force.

electrothermal arcjet

An electric thruster which heats the propellant to temperatures much higher than those obtainable in a chemical rocket by passing the propellant through an electrical arc discharge. Typical propellants include hydrazine, hydrogen, and ammonia. Hydrazine arcjet thrusters are currently the baseline for use on the Telstar 4 commercial communication satellite.

Hall effect

An effect resulting from the "side-ways" force which acts on a charged particle moving through a magnetic field. In solid conductors, this effect can be used to determine the sign of the charge carriers.

hollow cathode

A specific cathode design in which a low work function, high temperature material is placed inside a hollow tube which is partially closed at one end. A small flow of propellant gas through the cathode creates an internal plasma which facilitates emission of the electrons from the cathode surface and conduction of the electron current out of the cathode. State-of-the-art hollow cathodes are characterized by useful operating life times of greater than 10,000 hours.

ion engine

An electric thruster which first ionizes the propellant and then accelerates the resulting positive ions electrostatically. The accelerating electric field is established through the use of two or three closely spaced grids consisting of many thousands of carefully aligned apertures. Typical propellants include xenon, krypton, and argon.

larmor radius

Radius of the charged particle cyclotron motion around magnetic field lines of force.

power conditioning hardware

Power electronics hardware which converts the electric power obtained from the power source (photovoltaic solar arrays, for example) into the currents and voltages required to operate an electric thruster.

propellant efficiency

The fraction of propellant which is ionized and accelerated by an ion engine or a stationary plasma thruster.

specific impulse

There are two common definitions of specific impulse. The first is the ratio of thrust to the propellant mass flow rate. The unit of this specific impulse definition is *meters/second*. The electric propulsion community, however, often uses the related, but slightly different definition that the specific impulse is the ratio of thrust to the propellant weight flow rate (based on the value of the gravitational constant at the Earth's surface, $g = 9.8 \text{ m/s}^2$). In this case, the unit of specific impulse is *seconds*. The two definitions differ by the value of g .

thruster efficiency

Ratio of thrust power to the total input power for an electric thruster.

